[redacted]

N65-22569
(ACCESSION NUMBER)
31
(PAGES)
CR 62279
(NASA CR OR TMX OR AD NUMBER)
1
(THRU)
33
(CODE)
(CATEGORY)
FACILITY FORM 602

GPO PRICE $ ______

OTS PRICE(S) $ ______

Hard copy (HC) ______

Microfiche (MF) .50

Lockheed

MISSILES & SPACE COMPANY

A GROUP DIVISION OF LOCKHEED AIRCRAFT CORPORATION
SUNNYVALE, CALIFORNIA

5-07-24288

TECHNICAL REPORT
H-64-009
LMSC/HREC AO36515

LOCKHEED MISSILES & SPACE COMPANY

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A COMPARISON OF THE FLOW FIELDS ABOUT THE S-II STAGE OF THE SATURN V VEHICLE AND SPIKED HEMISPHERE CONFIGURATIONS WITH EMPHASIS ON THE S-II THRUST CONE HEATING RATES

Contract NAS8-11148

September 1964

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SUMMARY

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The flow field that results from interaction of the engine exhaust plumes with the external flow field about a vehicle when flying at high altitudes is examined and compared to the flow field about a spiked hemisphere configuration. Examination of both flow fields indicated that the inviscid portions for each configuration are similar, and that dissimilarities appear in the viscous separated flow fields. A discussion is presented in which the conclusion is reached that (1) single engine configurations are somewhat similar to spiked hemispheres when viscous flow fields are compared, because only a small quantity of the gas in the separated regions can be attributed to the exhaust plume; and that (2) multi-engine clusters result in a complex recirculation of exhaust gases that disturb the flow pattern in the separated flow region near the base, causing vast dissimilarities as compared to that found about spiked hemispheres.

A review of pertinent spiked hemisphere heat transfer data and a discussion of the theory of Chapman which predicts the average heat transfer rates in regions of separated laminar flow are included. Agreement between the data obtained on the hemisphere at the base of the spike and Chapman's theory is good. Expressions derived from geometrical considerations to relate flow separation angles and the spike length-hemisphere diameter ratio show excellent agreement with experiment and are used to supplement Chapman's theory.

Chapman's theory is used to determine the magnitude of the S-II stage-Saturn V thrust cone heating rates that would exist if external flow is considered without base bleed. These rates vary from 0.03 to 0.086 Btu/ft-sec depending upon plume diameter. Comparison with experimental data obtained for base bleed with no external flow reveal that the external flow field alone can produce heating rates comparable to those produced by exhaust gas recirculation. The external flow, therefore, cannot be ignored.

Author

FOREWORD

At the request of Mr. John Reardon for the Aero-Astrodynamic Laboratory, George C. Marshall Space Flight Center, the Huntsville Research & Engineering Center of Lockheed Missiles & Space Company, (LMSC/HREC), performed a study to determine the effect of the external flow field around the Saturn V vehicle on the base flow environment of the S-II stage. The usefulness for design purposes of base heating data obtained by Cornell Aeronautical Laboratory (CAL) was questioned because data obtained in CAL's short-duration base heating facility was obtained without simulation of the external flow effects. In particular, the heating rates occurring on the thrust cone structure after interstage separation appeared susceptible to external flow effects, making necessary an evaluation of the external flow field to ascertain a confidence level for applicability of the CAL data.

An experimental program was also considered in which the jet exhaust plume would be simulated by a properly shaped solid body attached to the base of the model. The usefulness of this program was also questioned.

An LMSC/HREC study was undertaken to determine: (1) if the external flow field appreciably affects the heating rates on the thrust cone structure, and (2) the usefulness and applicability of a wind tunnel program using a scaled model with a solid plume. This report contains the findings of this study.

CONTENTS

NOMENCLATURE

Symbols

C	Viscosity parameter in Chapman's Equation defined as $\left[T_w/T_e\right]^{0.5}\left[\frac{T_e+S}{T_w+S}\right]$
D	Diameter of hemisphere or plume
h	Enthalpy
h_{aw}	Adiabatic wall enthalpy
ℓ	Length of mixing layer
L	Length of spike
M_c	Equivalent cone Mach number
p	Pressure
q	Average heat transfer rate per unit area
q_{NS}	Heat transfer rate at stagnation point of hemisphere
S	Sutherland's constant (198.6)
T	Temperature
u	Velocity
x, y	Coordinates of Chapman's mixing layer analysis
γ	Ratio of specific heats
θ	Separation angle
σ	Prandtl number
ρ	Density
μ	Viscosity

Subscripts

e	Refers to static conditions at the outer edge of mixing layer or boundary layer
es	Refers to total conditions behind normal shock

∞	Refers to free-stream conditions
w	Refers to wall conditions

INTRODUCTION

During the ascent phase of such vehicles as the Saturn V, the base heating environment is complicated by the interaction of the recirculating engine exhaust gases and the external flow about the vehicle. At high altitudes, engine exhaust gases expand to form large plumes because the jet-static-to-ambient-static pressure ratio becomes large. As the size of the plume increases, it interferes with local flow about the body. At the intersection of the plume streamline and the external flow streamline, both streamlines deflect and flow in the same direction. Since the flow is supersonic, a shock wave forms (with accompanying pressure rise) to deflect the inviscid streamlines that lie outside the intersecting streamlines. However, the shock does not penetrate the viscous boundary layer in the sense that a sharp increase in pressure occurs. The pressure rise in the boundary layer increases slowly over a finite distance, setting up an adverse pressure gradient. As the altitude increases and the plume size increases, the adverse pressure gradient increases to a level that can no longer be negotiated by the boundary layer flow. The boundary layer then separates from the vehicle. As the flow separates, a shock forms at the separation point, turning the inviscid streamlines away from the vehicle, thus requiring less turning when the plume intersection region is reached. At high enough altitudes, the Reynolds number decreases to the point that the boundary layer becomes entirely laminar. Because laminar boundary layers separate at much lower adverse pressure gradients than do turbulent boundary layers, the separation point occurs much earlier on the vehicle. Finally, the separation point occurs at the nose of the vehicle. There appears to be some similiarity between the inviscid flow field about the vehicle and the flow field formed by attaching a long, slender rod or "spike" to the stagnation point of a blunt body.

When the S-II stage of the Saturn V vehicle is operating, (a schematic of the S-II base is shown in Figure 1) the external flow field is characterized by the high velocity ($M_{\infty} > 9.0$), high altitude (Alt. 275,000 feet) flight regime.

Distinguishing traits of the flow field that are common to this phase of the flight are:

1. Large jet plumes resulting in boundary layer separation over a considerable portion of the vehicle.
2. A very thick, but extremely low-density, laminar mixing layer.
3. Strong interactions between the viscous boundary or mixing layer and the inviscid shock layer.
4. High enthalpies, resulting from high velocities in a rarified atmosphere.

Although these low density aspects complicate the flow field, continuum flow theories are sufficiently accurate to determine the inviscid flow field. The nature of the viscous flow field, especially that in close proximity to the thrust cone structure, can only be surmised at this time.

Assuming that the flow field about the thrust cone is similar to the flow field near the base of the spike of the spiked hemisphere configuration, the thrust cone heating rates can be predicted by Chapman's theory (Reference 1). Comparisons of experimental heat transfer rates on a spiked hemisphere with Chapman's theory are included in this report to show the theory's applicability for this type of flow problem. Predictions of the heating rates on the thrust cone by use of Chapman's theory are also included. A comparison between the Cornell Aeronautical Laboratory data (which were obtained with no external flow) and Chapman's theory for no-base-bleed provides an indication of whether or not the external flow field contributes enough to thrust cone heating to be significant.

An accurate determination of the thrust cone flow environment and heating rates would require an extensive experimental investigation. Various test programs, with particular emphasis on spiked-hemisphere tests, will be discussed for their merits. Recommendations are made concerning the course of action that should be taken to determine the S-II stage base flow environment.

DISCUSSION

S-II Flow Field

For purposes of calculation and discussion, the initial altitude and Mach number of the S-II stage is assumed to be 275,000 feet and 9.4, respectively. At this altitude, the free stream kinematic viscosity is approximately 13 square feet per second and the mean free path is approximately 3.25 inches. The unit Reynolds number is found to be approximately 600 per foot and Reynolds number based on vehicle length is on the order of 10^5. This indicates that the flow over the vehicle will be completely laminar. However, with a mean free path of 3.25 inches at this altitude, the use of Reynolds number criteria to describe boundary layer conditions may not be feasible. Since the magnitude of the Knudsen number (i. e. mean free path divided by a characteristic length) reflects the flow regime, the type of flow field can be determined. If the vehicle length is used as the characteristic length, the Knudsen number is sufficiently small that continuum flow will exist; however, the nature of the problem is such that the vehicle length may not be the proper characteristic length. The problem is a local boundary layer one because heat transfer rates in a separated region are desired; therefore, the Knudsen number is more realistic if based either on boundary layer thickness or on the diameter of the S-II stage. Since the boundary layer would be several feet thick and somewhat more dense than the undisturbed air, the Knudsen number based on local mean free path and boundary layer thickness would probably be of the same order of magnitude as that based on free stream conditions and vehicle diameter. This value is approximately 0.001. The viscous layer regime of low density flow is generally considered to exist when the value of Knudsen number exceeds 0.01. Based on vehicle diameter and free stream values of mean free path, this value corresponds to an altitude of approximately 320,000 feet. Therefore, the initial portion of the S-II flight will be in continuum flow regime; however, this continuum flow regime is so near transition that second order effects of vorticity, slip, temperature jump, curvature (both longitudinal and transverse as related to solution of thick boundary layers equations) and thick boundary layers (large values of δ^*/R that effectively change the inviscid body shape) may become significant. For this analysis, these second order effects are ignored.

At an altitude of 275,000 feet, the jet exhaust gases expand to form a plume that is the size of several vehicle diameters. With the combination of this large plume and laminar boundary layer, the boundary layer is separated over most of the vehicle's length. The flow pattern in the separated region is complex and for the most part undefined. However, the flow pattern (see Figure 2) can be detailed somewhat by surmising both the inviscid flow field and the flow mechanics that must occur in the region of the intersection of the external flow with the plume. If we assume that the S-II base consists of one jet rather than a five-jet cluster so that symmetry exists, the inviscid flow field can be approximated. This procedure, though iterative, is simple. An inviscid external flow streamline of known Mach number and direction intersects with an inviscid, exhaust gas streamline of known Mach number and direction. At the point of intersection, oblique shocks are formed which are strong enough to turn both streams to the same direction. Unfortunately, the original streamlines are curved, yielding numerous answers which are functions of the physical location of their intersection. To circumvent this, some knowledge of the viscous flow is necessary. Since laminar boundary layers are separated easily by small adverse pressure gradients, the static pressure rise must be small across the shock at the intersection. To maintain a small pressure rise across this shock, the streamlines can experience only a small directional change. Therefore, the streamlines must intersect at a point where the angle between them is small. Also, the angle at which the laminar boundary layer separates must be small, or approximately at free-stream Mach angle. Subsequently, an initial assumption on which to base the inviscid flow approximation for the S-II is to assume that the separation point occurs at the nose of the vehicle, and reattachment or intersection with the plume occurs near the point where the shock layer is tangent to the plume. If the separation angle is nearly equal to the Mach angle, separation may occur aft of the nose. Naturally, the plume shape must be known in order to determine the intersection point. A good approximation to use in determining plume shape is to assume that the plume expands into quiescent air, which has a static pressure equal to the local surface pressure on a cone whose semi-vertex angle is equivalent to the separation angle. Applying this criterion to the S-II, the separation point should be at the nose for plume diameters greater than two body diameters; consequently, separated flow should exist over the entire vehicle at the flight condition under consideration.

The preceding discussion has assumed that little or no knowledge of the viscous flow was known. However, the basic theory postulated by Chapman et al (Reference 2) and Korst et al (Reference 3) can be used to describe the general pattern of the viscous flow. A dividing streamline can be defined, then, as that streamline in the mixing layer that, when allowed to stagnate, has a pressure equal to the static pressure at reattachment. Then any streamlines that lie outside the region formed by the body and the dividing streamline have enough energy to negotiate the pressure rise at reattachment and to pass downstream. However, any streamline that is inside this region will have insufficient energy to negotiate the pressure rise and will be recirculated in the so-called "dead air region." The scavaging action of the mixing layer maintains a mass flow balance in this dead air region. In this case, however, two gas streams are intersecting; therefore, two dividing streamlines, one in each stream, will exist. Both dividing streamlines must stagnate to the same pressure since they terminate at the point of their intersection. Futhermore, the static pressure in the dead air region and throughout the mixing layer is assumed to be constant. (This assumption is poor in axisymmetric flow since a static pressure rise along the streamline will exist; however, this assumption is relatively unimportant to the conclusions sought.) The pressure rise obtained by stagnating the laminar dividing streamline is small in comparison to the rise that would exist for the plume dividing streamline since the mixing layer in the plume is undoubtedly turbulent. However, the stagnation and static pressure on the laminar dividing streamline must equal the stagnation and static pressure on the turbulent dividing streamline. Since the laminar mixing layer will intersect the plume at a location dictated by some minimum pressure rise, the dividing streamline of the plume must adjust to this pressure rise. Therefore, the dividing streamline in the plume will have a lower energy level, as dictated by the laminar pressure rise, than is normal for purely turbulent flow. This has the effect of displacing the dividing streamline toward the dead air region. Consequently, a much smaller portion of the plume gas will be recirculated when the external mixing layer is laminar than when it is turbulent. For the present case, this indicates that most of the gas recirculated by the intersection of these two streams is obtained from the external stream. If we consider this intersection only, and assume that there is only a single engine exhaust (this neglects the effect of plume intersections with each other that would recirculate a considerable amount of exhaust gases into the separated region), this flow field will be similar to that about a spiked hemisphere.

The discussion thus far has been limited to flow-field interaction of external flow with a plume issuing from a single jet. The real flow field about the base of the S-II is more complicated. There is a jet plume interaction between each pair of the five-jet cluster, and exhaust gases that flow into the separated region about the vehicle are recirculated. Changes in direction and velocity of this recirculating gas that occur as a result of engine gimbaling could significantly alter the flow pattern and, consequently, the heating rates over the thrust cone. Other complicating factors in determining the flow pattern over the thrust cone result from the lack of symmetry of the flow field. The exhaust plume will be non-axisymmetric to some extent because of the configuration of the jet cluster; consequently, the intersection of the external stream with the plume will be non-axisymmetric. Local cross flow that may exist could significantly alter the separated flow pattern at some conditions. Fortunately, the S-II is at a sufficiently high altitude that the effect of this asymmetry will be slight. The effect of asymmetry decreases as altitude increases. Another factor leading to asymmetry of the flow field is the flight altitude of the vehicle itself. The angle-of-attack throughout the flight is sufficient to affect the heating rates on the thrust cone. This angle-of-attack effect is relatively undefined. It adds complexity to a problem that is already extremely complicated at zero angle-of-attack.

Spiked Hemisphere Flow Field

The similiarity of the flow field about the S-II to that about a spiked hemisphere was noted during the course of the study. The hemisphere forms a restriction to the local flow over the spike in much the same manner that the plume does to a vehicle. The mechanics of the flow separation in both cases are similar, and the inviscid flow field that results is similar. Therefore, it appeared likely that the heating rates on the S-II thrust cone resulting from external flow could be duplicated by a spiked hemisphere configuration. However, the flow pattern in the separated region is dissimilar when one compares the pattern shown for the S-II (Figure 2) and the spiked hemisphere (Figure 3). Gas bleed into the separated region of the spiked hemisphere must be included to more nearly simulate the flow pattern on the S-II. Unless the direction, velocity, and mass flow rate of this gas injected at the base of the spike are similar to that in the S-II base flow pattern, the desired simulation is not obtained. Therefore, simulation of the flow pattern over the thrust cone by use of a spiked hemisphere, including gas bleed, is impossible until the S-II flow field is accurately defined.

The flow pattern of the separated flow region of the spiked hemisphere is well described by Chapman in Reference 1. Therefore, Chapman's theory to predict the average heat transfer rates in separated flow can be expected to yield good results when it is applied to a spiked hemisphere. The ability of Chapman's theory to predict these heating rates can be assessed by comparing them with existing experimental data. Unfortunately, agreement of Chapman's theory with spiked hemisphere data does not indicate that the theory can be used to predict the thrust-cone heating rates. If it is found that the flow fields are similar, Chapman's theory can be used in lieu of performing spiked-hemisphere experiments to determine the heating rates.

Spiked Hemisphere Experiments

Tests on spiked hemisphere configurations were conducted at supersonic velocities as early as 1951. At that time, emphasis was placed on finding a method of reducing drag on a blunt body. As knowledge of the mechanics of separated flow increased, interest in the spiked blunt body increased, especially at hypersonic velocities. Much of this experimental work was performed by Bogdonoff at Princeton University in the late fifties. Some of this work is included by Bogdonoff in Reference 4, which presents a survey of the experimental work performed in hypersonic separated boundary layers. Experiments that were performed measured only the total heat transfer rate to the body rather than to local distributions. Crawford (Reference 5) presents one of the most comprehensive tests on spiked hemispheres that have been made. He obtained heat transfer and pressure distribution data on a spiked hemisphere at Mach 6.8 for various Reynolds numbers and ratios of spike length to body diameter. Zakkay (Reference 6) presents similar data at Mach 7.9 for a spiked body. Here the body is a spherically tipped cone, having a cone half angle of 50 degrees. Since he measured the total heat transfer rate to be greater for the spiked body than for an unspiked body, the flow in the mixing layer must experience transition. The only data in References 4, 5 and 6 that are entirely laminar and can be used to substantiate Chapman's theory is Crawford's low Reynolds number data. In the following discussions reference to Crawford's data is intended to mean reference only to his minimum Reynolds number data.

Crawford measured local heat transfer rates to the surface of the hemisphere. His results for a point on the hemisphere near the spike-hemisphere junction are presented in Figure 4. The use of L/D as an independent variable in presenting the heat transfer rates has become standard; however, the use of the separation angle, θ, as the parameter is quite interesting. The variation of q/q_{NS} with θ is presented in Figure 5.

The heat transfer rates are found to be a function of the separation angle. This is expected because it has long been known from experimental observations that the static pressure in the separated regions can be approximated by assuming the static pressure to be equal to local surface pressure on a cone of half angle θ. Consequently, one would also expect that the increase of q/q_{NS} as θ increases can be predicted by theory. Use of Chapman's theory (Reference 1) utilizing the density and velocity on a cone of half angle θ and dividing by Fay and Riddell's theory (Reference 7) for a hemisphere, assuming D is known, should at least predict the trend, but perhaps not the magnitude of the ratio q/q_{NS} as θ varies.

Separated Flow Geometry Relations

As shown in Figure 6, as L/D decreases, the separation angle, θ, can be expressed as $\theta = \sin^{-1} \left(\frac{D}{2L+D} \right)$ which infers that the flow separates at the spike tip and reattaches tangentially to the hemisphere. Actually, the reason for such a relationship results from forcing the separation point to occur at the tip of the spike. When the spike becomes sufficiently long that separation occurs "naturally," (i. e. a minimum adverse pressure gradient is required for separation which would be constant for any particular Mach number) then additional increases in spike length will not affect the separated flow field except for the natural increase in boundary layer thickness that occurs before separation. For this case the relationship $\theta = \sin^{-1} \frac{1}{M_\infty}$ should yield fair results.

It is interesting to compare the separation angles measured by Crawford on the spiked hemisphere with those found by Falanga et al (Reference 8) on a cone-cylinder-flare configuration with a jet plume issuing from its base. At a free stream Mach number of 9.65 and a jet-to-free-stream static pressure ratio of approximately 1000, the separation angle was 12 degrees. The value of L/D appears to be approximately 1.5 — 2.0, which compares well with Crawford's data. As the plume size decreases, the separation angle decreases, which of course, is equivalent to a L/D increase. The mechanics of separation are,therefore, similar. Unfortunately, data obtained in the tests reported in Reference 8 were entirely photographic. Thus, no pressure or heating rate comparison can be made with spiked hemisphere data.

Chapman's Theory

In 1956, Chapman published a theory for determining an average heat transfer rate in cavity-type separated flow for cases in which the boundary layer remained laminar through the reattachment point. In developing this theory Chapman made several assumptions and their consequence as pertains to either the results of a wind tunnel test or to the flight conditions of the S-II are:

1. In order to solve the differential equations of momentum, continuity, and energy for viscous flow within a relatively thin, constant pressure laminar mixing layer of length ℓ, Chapman assumes the following boundary conditions: $u(x,\infty) = u_e$; $h(x,\infty) = h_e$; $u(x,-\infty) = 0$; and $h(x,-\infty) = h_w$. These conditions infer that the

outer boundary of the mixing layer is at $y = \infty$ and the x-component of velocity u and enthalpy h are constant since pressure is assumed constant and equivalent to the local inviscid values. Similarly, the inner boundary of the mixing layer is the wall which is located at $y = -\infty$. Use of $y = -\infty$ as a boundary condition means that the mixing layer characteristics are determined as though the wall has no effect on the mixing layer. Chapman states that his assumption is good if the thickness of the mixing layer is small as compared to the dead air region. At the low-density high-enthalpy flight conditions that exist for the S-II, this assumption may not be as valid as compared to the low enthalpy wind tunnel test on a spiked hemisphere.

2. Prandtl number is assumed to be constant throughout the mixing layer. This assumption is quite common and it would not be expected to have a significant effect in this case.

3. The gas is assumed to obey the perfect gas law with enthalpy being a function of temperature only. Some effect would be found for this case when the enthalpy in the mixing layer is quite high.

4. The viscosity variation across the mixing layer is similar to the static temperature variation and is written as $\mu/\mu_e = C\ T/T_e$. This is a typical assumption.

5. The mixing layer thickness at separation is "0" or negligible when compared to ℓ. As long as the separation point occurs at the nose this assumption will be valid.

Actually, the only assumptions that make Chapman's theory somewhat inaccurate for this situation are those concerning mixing layer thickness. The ability of this theory to predict heat transfer rates when the mixing layer becomes thick has not been evaluated. It is felt, however, that an "order of magnitude" answer can be obtained if this theory is used with the assumption that the S-II exhaust plume is a solid boundary with no recirculating exhaust gases.

Chapman's equation for the average heat transfer rate for axisymmetric separated flow with no gas injection and Prandtl number equal 0.72 can be written as follows:

$$q = 0.529 \sqrt{\frac{\rho_e u_e \mu_e}{\ell}} \sqrt{C} \; (h_{aw} - h_w)$$

Comparison of Chapman's Theory with Experimental Data

An analytical expression can be derived from Chapman's theory and Fay and Riddell's theory that will directly yield the ratio q/q_{NS}. This expression is

$$\frac{q}{q_{NS}} = \left[\frac{R}{\ell}\right]^{0.5} \frac{\sigma^{0.6} K}{0.763} \left[\frac{p_e/p_\infty}{p_{es}/p_\infty}\right]^{0.5} \left[\frac{T_e/T_\infty}{T_{es}/T_\infty}\right]^{0.25} \left[M_c\right]^{0.5}$$

$$\left[\frac{T_{es} + S}{T_w + S}\right]^{0.4} \left[\frac{T_w}{T_{es}}\right]^{0.2} \left[\frac{\gamma}{2(1 - p_\infty/p_{es})}\right]^{0.25} \left[\frac{r - (1-r)\frac{h_e}{h_{es}} - \frac{h_w}{h_{es}}}{1 - \frac{h_w}{h_{es}}}\right]$$

where K and r are functions of the Prandtl number σ; for $\sigma = 0.72$ and $\gamma = 1.4$, the expression becomes

$$\frac{q}{q_{NS}} = 0.570 \left[\frac{R}{\ell}\right]^{0.5} \left[\frac{p_e/p_\infty}{p_{es}/p_\infty}\right]^{0.5} \left[\frac{T_e/T_\infty}{T_{es}/T_\infty}\right]^{0.25} \left[M_c\right]^{0.5} \left[\frac{T_{es} + S}{T_w + S}\right]^{0.4}$$

$$\left[\frac{T_w}{T_{es}}\right]^{0.2} \left[\frac{0.7}{1 - p_\infty/p_{es}}\right]^{0.25} \left[\frac{0.85 - .15\frac{h_e}{h_{es}} - \frac{h_w}{h_{es}}}{1 - \frac{h_w}{h_{es}}}\right]$$

If the assumption is made that $\frac{h_e}{h_{es}} = \frac{T_e}{T_{es}}$ and $\frac{h_w}{h_{es}} = \frac{T_w}{T_{es}}$; then the ratio q/q_{NS} can be determined by knowing only M_∞, θ, $\sqrt{R/\ell}$, and the ratio of T_w/T_{es} desired. The term $\sqrt{R/\ell}$ can also be expressed in terms of θ. Since the relationship $\theta = \sin^{-1}\left[\frac{D}{2L+D}\right]$ agrees with experimental data as shown in Figure 5, then

$$\sqrt{R/\ell} = \sqrt{\tan\theta}$$

This expression was used in calculating the theoretical q/q_{NS} versus θ curve presented in Figure 4. Comparisons with Crawford's data indicate that the theory tends to under-predict the heat transfer rates on the hemisphere; however, one would expect that the rates on the spike near its base will be lower than those on the hemisphere. Therefore, the theory should yield good results.

Heat Rate Prediction for the S-II

Calculating the values of q/q_{NS} and q_{NS} for conditions at M = 9.4 at an altitude of 275,000 feet, the variation of q can be presented as a function of D for various altitudes (see Figure 7). For the case of the large plume emanating from the S-II, the value of L remains constant while D increases. Thus, an increase in altitude is equivalent to a decrease in L/D. If the plume diameter is known, then an estimate of q can be obtained from Figure 7. It must be pointed out that in the experiments of Crawford, the ratio of the diameter of the spike (D_s) to diameter of the body (D) was maintained at 0.10. The effect of this ratio on the heat transfer rates is unknown. However, for values of $D_s/D \leq 0.10$ it can be expected to be slight. This sets the minimum plume diameter at 330 feet, which is probably about half the actual diameter of the plume at an altitude of 275,000 feet. With this being the case, the ratio of L/D would be less than 0.5 and θ greater than 30 degrees. Since the variation of q with D is relatively insensitive to D for values of D greater than 600 feet, q primarily becomes a function of altitude.

The estimated variation of q with time for the S-II thrust cone (holding diameter constant at 800 feet) is presented in Figure 8. The increasing trend of q with time (12-24 seconds) is caused by the rapid increase in total enthalpy that occurs as a result of increasing ambient temperature and decreasing ambient pressure as altitude increases. The accuracy of this method at the high altitudes must be questioned because of the overpowering low density effects.

It is interesting to compare maximum heating rates estimated in Reference 9 (obtained from the results of the heat transfer tests made at CAL without the effect of external flow) with those calculated herein, in which no recirculation of exhaust gases is assumed. Predictions of the heating rates for the 210-inch heat shield vary from 0.08 to 0.50 Btu/ft^2-sec over the thrust cone with the maximum rate occurring at S-II station 126. Maximum rates on the thrust cone with the 256-inch heat shield in place vary from 0.05 to 0.20 Btu/ft^2-sec. The calculated maximum average rate that results from external flow alone is 0.086 Btu/ft^2-sec. For the 256-inch heat shield case, the external flow average heating rate is almost half of the predicted heating rate. The effect of external flow on the thrust cone heating rates can obviously be quite significant.

CONCLUSIONS

The preceding discussion compared the flow field that exists about the Saturn V vehicle during operation of the S-II stage with the flow field that forms about a spiked hemisphere configuration. The use of a spiked hemisphere configuration to determine the effect that external flow will have on the thrust cone heating rates is unrealistic because:

1. Recirculation of the plume gas into the separated region (although only a small amount of the plume gas will recirculate) is not duplicated on a spiked hemisphere.
2. The gas flow from the S-II heat shield is not simulated.
3. Reattachment of the separated boundary layer occurs on a hot gas plume, not a cold wall. (This may affect the recirculation mechanics.)

Although these three reasons are enough to discourage designing an experimental program based on use of the spiked hemisphere, there are additional reasons. They are:

1. Empirical predictions of the heating rates can be made on experimental data which already exists. (This assumes that the flow field is independent of the shape of the spike base so that the heat transfer rates on the thrust cone are the same as those near the base of a cylindrical spike.)
2. Chapman's method is available for use in predicting the heating rates in the area of interest.
3. Although local heat transfer rates on the spike have not been measured, these rates will undoubtedly be lower than those chosen for this comparison. Therefore, at the spike hemisphere junction and forward onto the spike, Chapman's predictions should be conservative.

It is interesting to note that relatively simple relationships can predict the separation angle for known values of L/D. These relationships are expected to be independent of Mach number as long as the mixing layer is completely laminar. Additional spiked hemisphere, or spiked blunt-body data would be beneficial for use in perfecting the relationships and methods used in this analysis. However, these additional data would not be especially helpful in solving the S-II problem.

The S-II problem is indeed complex. The probability of solving it analytically is poor, principally because of lack of knowledge of the separated flow pattern. The heat transfer rates presented here for external flow with no base bleed are of the same order of magnitude as the rates of heat transfer measured at CAL for the simulated 256-inch heat shield in place with no external flow. We, therefore, conclude that the external flow field cannot be ignored. To evaluate thoroughly the effect that external flow has on the base heating environment, experimental data are a necessity. The experiments must be so designed that the effect of each parameter can be evaluated separately; otherwise, the data obtained will be applicable only to one base configuration. Since it is likely that this problem will continue to exist on future vehicles, a parametric study should be financially attractive.

RECOMMENDATIONS

An experimental program is recommended to evaluate thoroughly the effect of external flow on the thrust cone heat rates. This experimental program should be somewhat broad in nature in order to determine the contribution of each flow source on the heating rates. To attain this, the following experiments are necessary:

1. A hypersonic wind tunnel test of a model so designed that a large jet plume can be formed by expansion of a gas through a single nozzle in its base. Variations in pressure distributions and heating rates can be measured on the model for various separation flow fields which can be set up by varying either or both the operating pressure and the temperature of the tunnel and jet. This test will investigate the external flow-plume intersection effect only.

2. A simple academic test to determine the basic viscous effects and to evaluate the ability of various analytical methods (References 1, 2 and 3) in determining the mass flow reversed by intersection of two nozzle exhaust plumes. This test could be accomplished in a vacuum chamber with no external flow simulation. Various base configurations should be investigated.

3. A hypersonic wind tunnel test similar to recommended Test 1, except that a typical S-II base configuration is used. This would allow a fully simulated flow field to be studied and the results obtained should be directly applicable to the S-II problem.

Test 3 is the only one directly applicable to the S-II. However, when basic changes in base configuration occur, the results of Test 3 are almost useless; whereas if all three tests are performed, the resulting information should provide all the data required for predicting the heating rates on the new configurations. Therefore, it is strongly recommended that a parametric-type experimental program, involving all three phases, be initiated and carried to conclusion.

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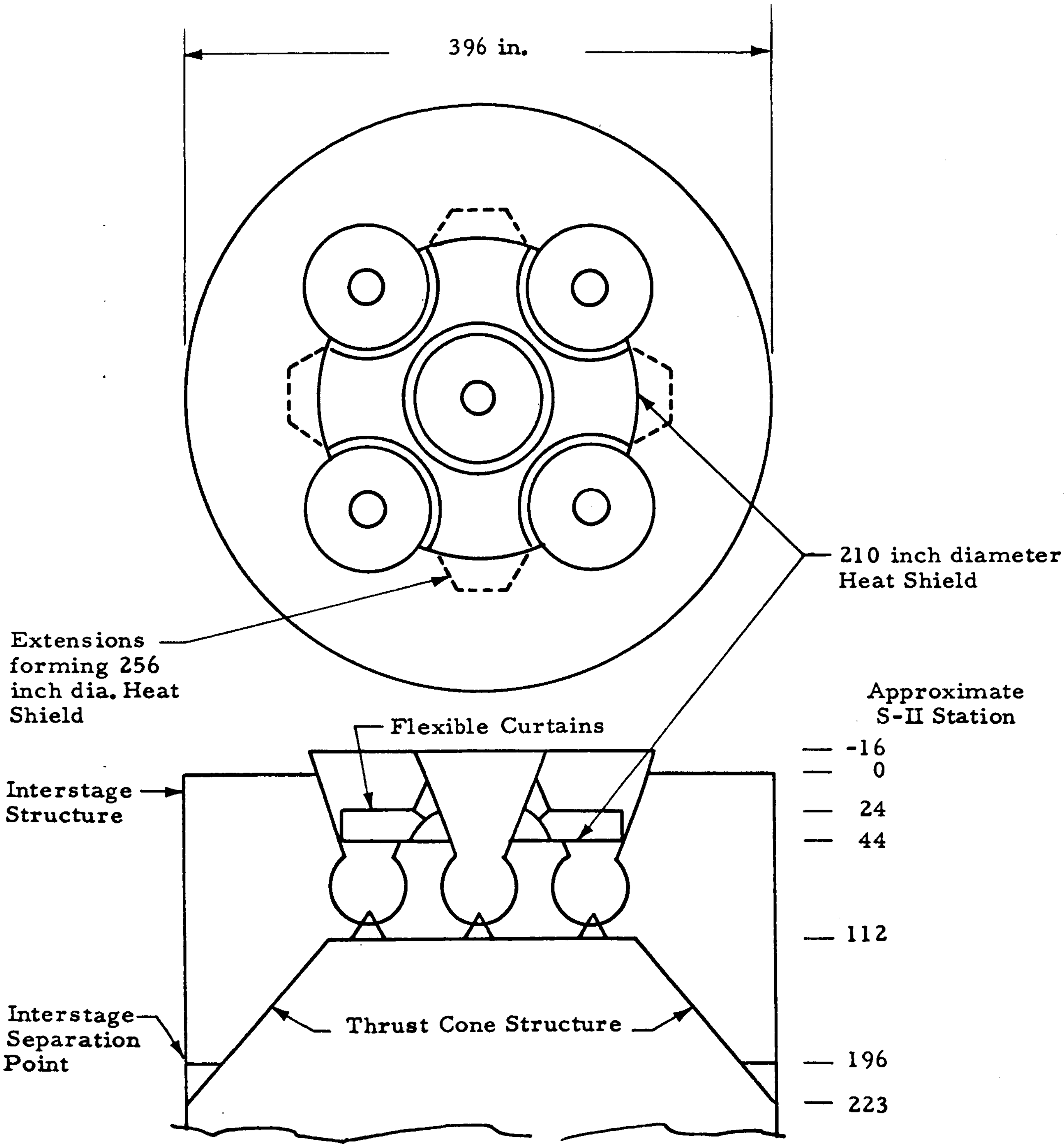

Figure 1 - Saturn S-II Base Configuration

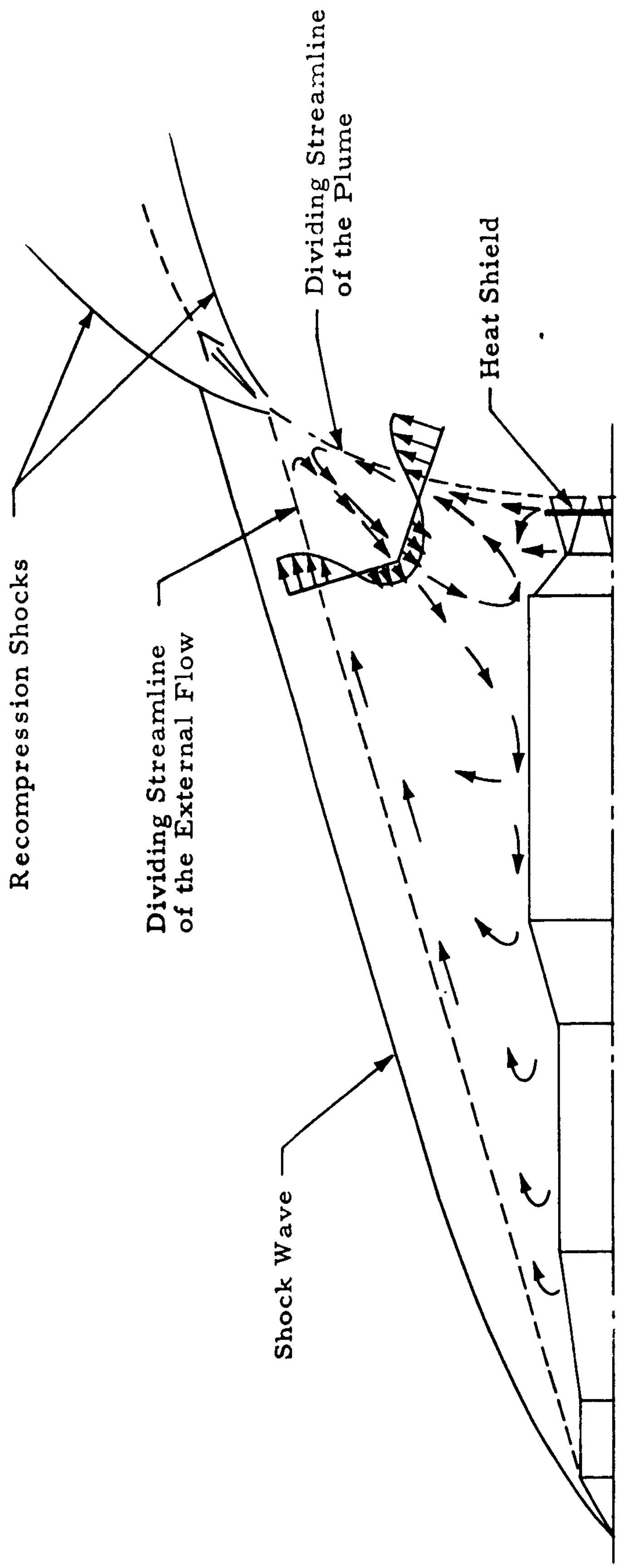

Figure 2 - Saturn V Flow Field

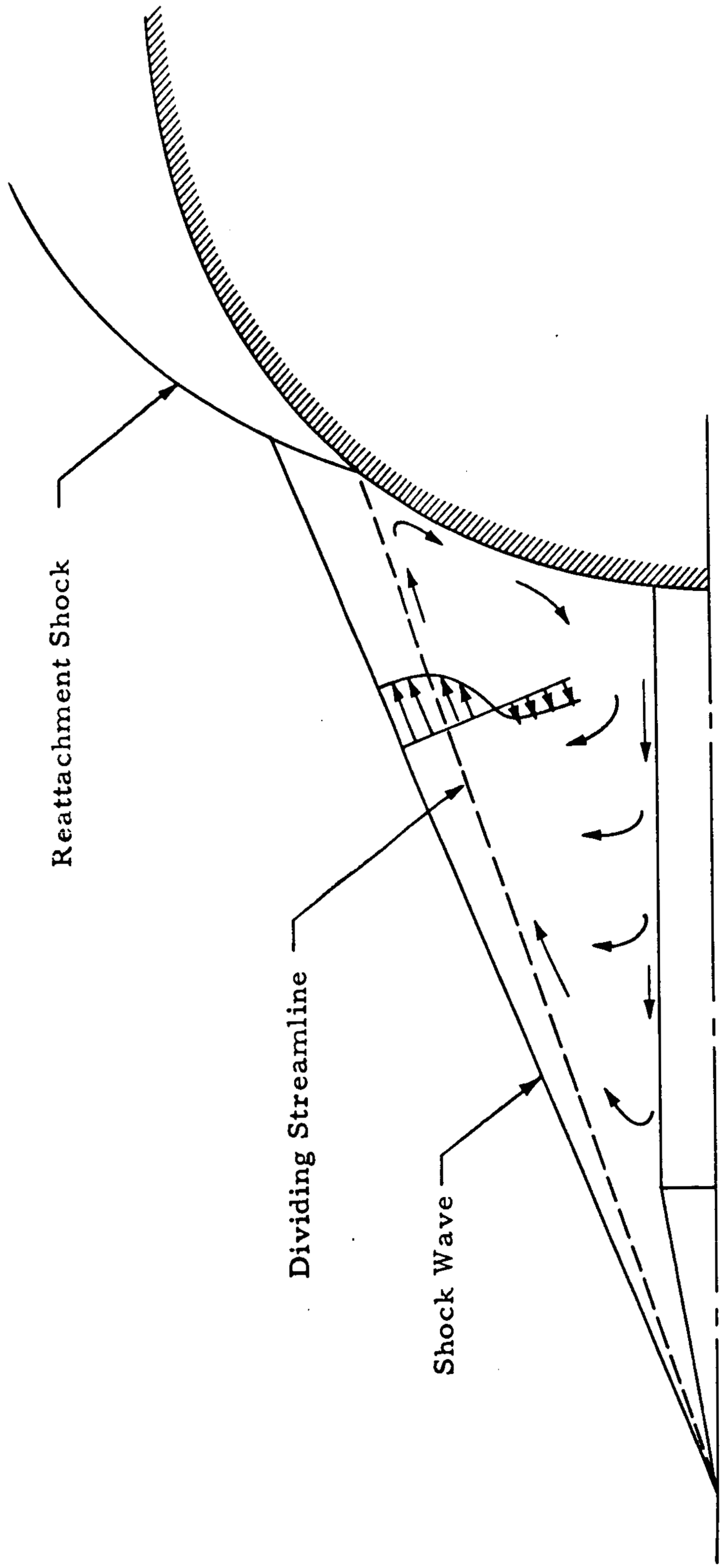

Figure 3 - Spiked Hemisphere Flow Field

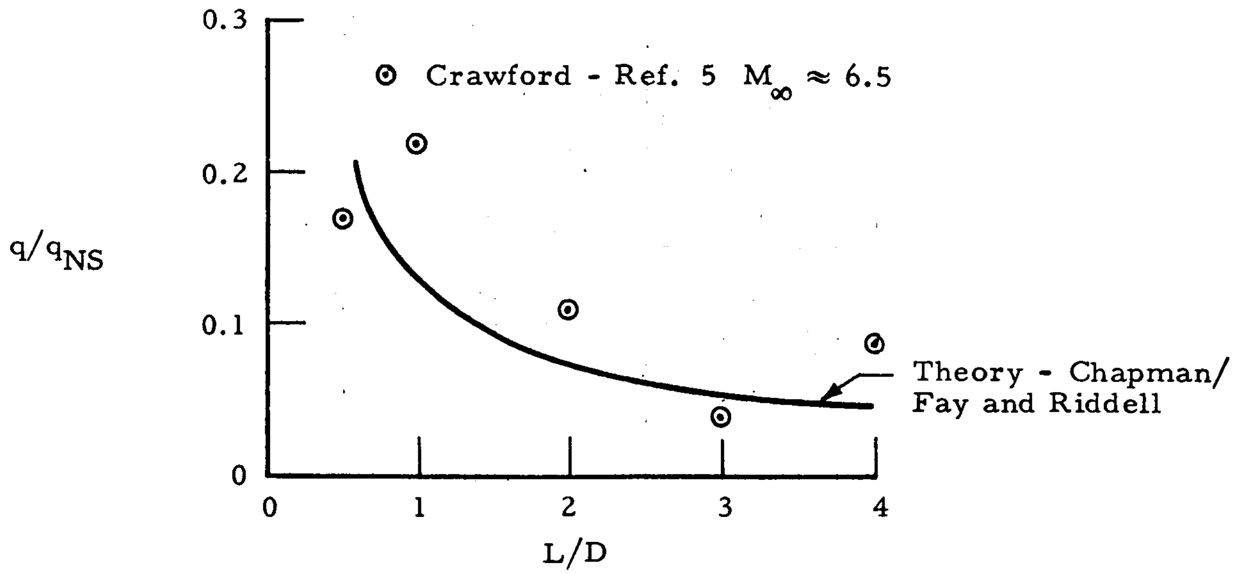

Figure 4 - Variation of the Heat Transfer Rates Near the Spike Hemisphere Junction with L/D

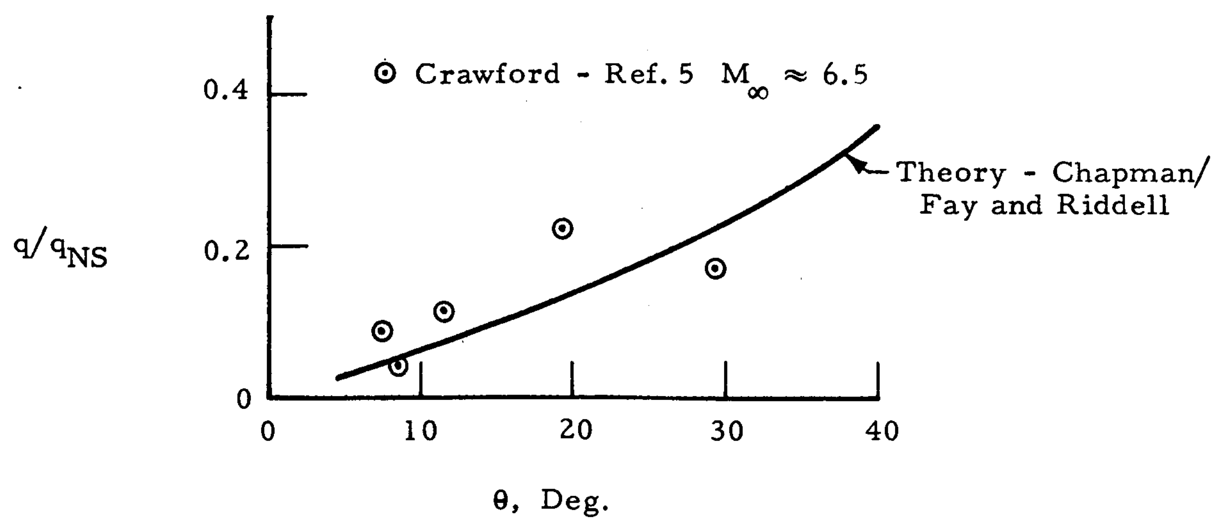

Figure 5 - Variation of the Heat Transfer Rates Near the Spike Hemisphere Junction with Separation Angle

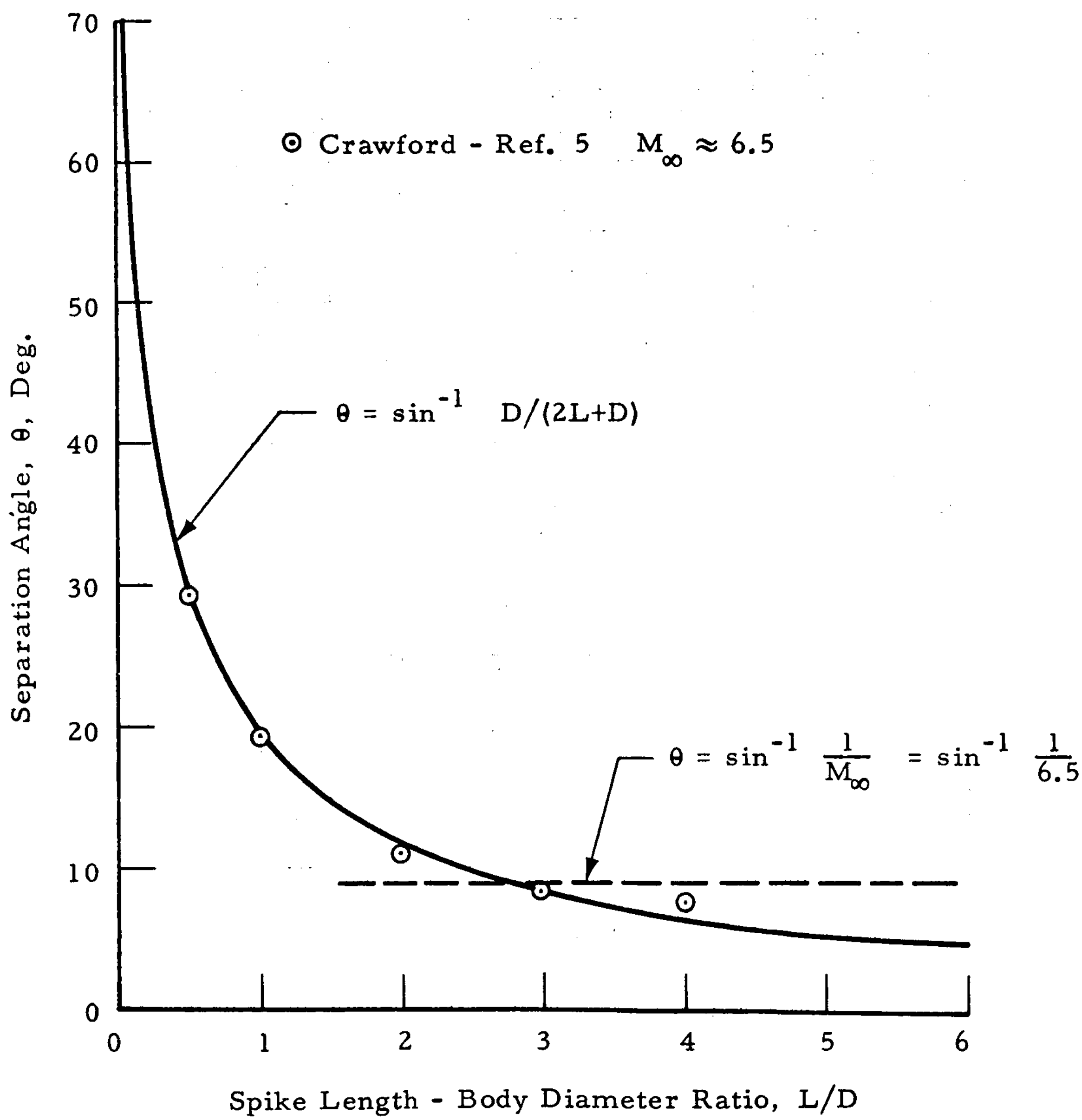

Figure 6 - Variation of Separation Angle with L/D for a Spiked Hemisphere in Laminar Flow

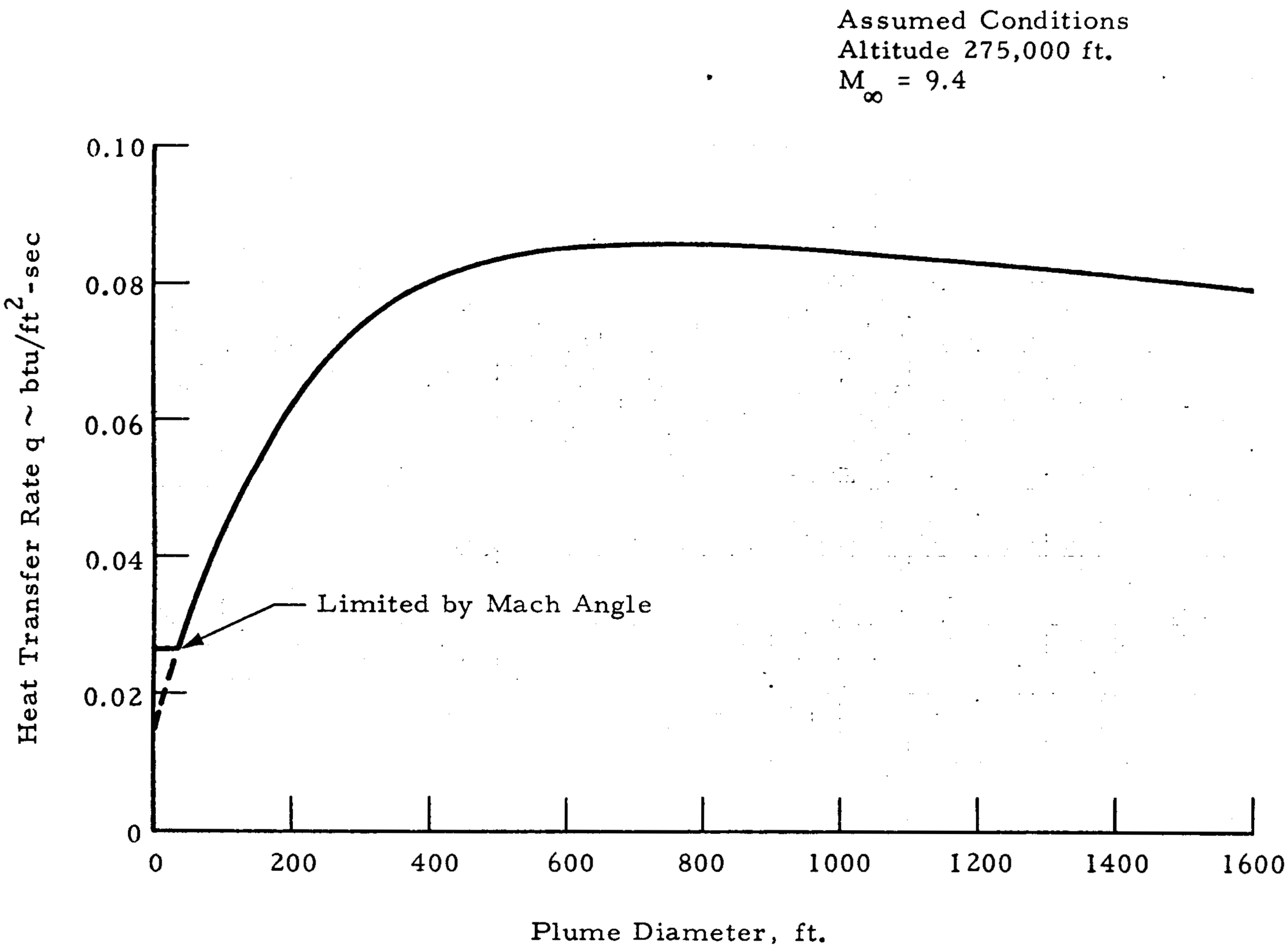

Figure 7 - Estimated Heat Transfer Rates on the S-II Thrust Cone Resulting From External Flow using Chapman's Theory

Note: Rates are based on external flow considerations only.

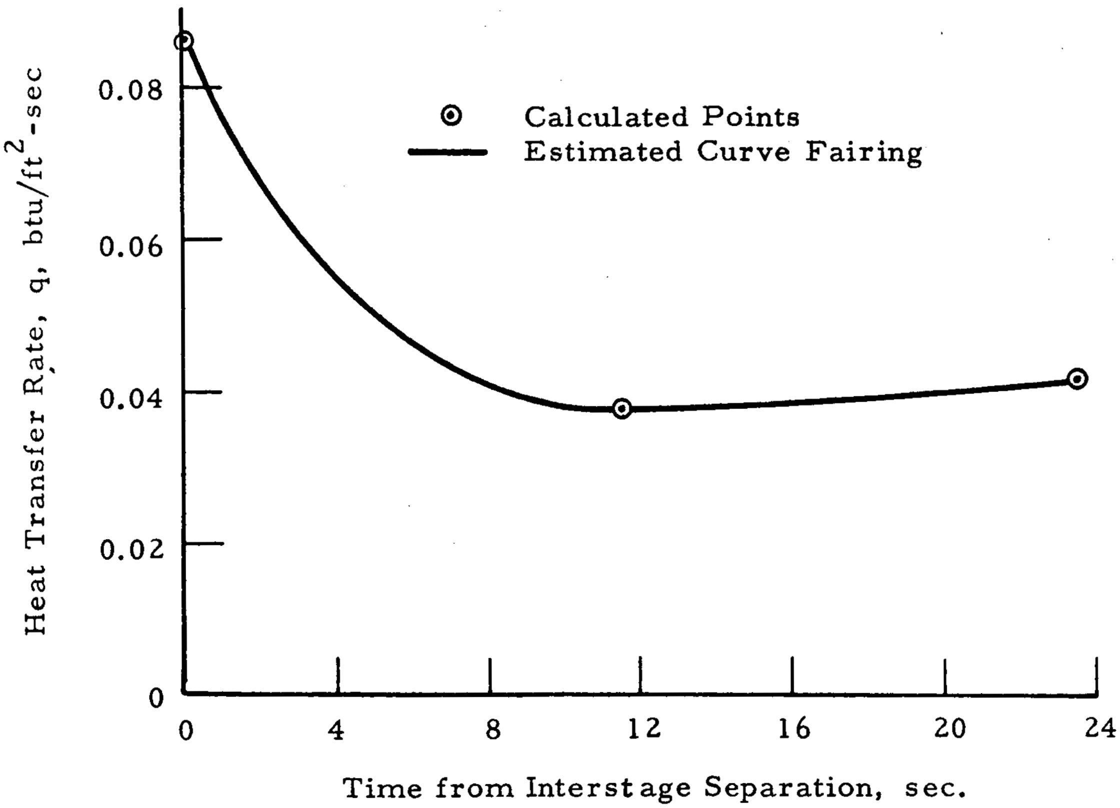

Figure 8 - Variation of Maximum Heating Rates with Time for the S-II Thrust Cone as Estimated by Chapman's Theory